U. S. ARMY
TRANSPORTATION RESEARCH COMMAND
FORT EUSTIS, VIRGINIA

# EXPERIMENTAL RESEARCH

## CH-21A HELICOPTER AIRFRAME DEFORMATION UNDER A DYNAMIC CRASH CONDITION

One of a Series of Reports
Pertaining to the Dynamic Crash Test of a U. S. Army CH-21A Helicopter

January 1964

Contract DA 44-177-AMC-888(T) TRECOM Technical Report 63-77



prepared by :

AVIATION SAFETY ENGINEERING AND RESEARCH
PHOENIX, ARIZONA
A DIVISION OF
FLIGHT SAFETY FOUNDATION, INC.
NEW YORK, NEW YORK

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This report was prepared by Aviation Safety Engineering and Research (AVSER), a division of the Flight Safety Foundation, Inc., under the terms of Contract DA 44-177-AMC-888(T). Views expressed in the report have not been reviewed or approved by the Department of the Army; however, conclusions and recommendations contained therein are concurred in by this Command.

Where applicable, the recommendations contained in this report will be incorporated in the design specifications of future U. S. Army aircraft.

JERRY L. REED
Project Engineer

WOODBURY JOHNSON, Lt Col, TC
Group Leader
Human Factors & Survivability Group

APPROVED.

FOR THE COMMANDER:

LARRY M. HEWIN
Technical Director

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# CH-21A HELICOPTER AIRFRAME DEFORMATION UNDER A DYNAMIC CRASH CONDITION

One of a Series of Reports Pertaining to the Dynamic Crash Test of a U. S. Army CH-21A Helicopter

Technical Report
AvSER 63-3

Prepared by
Aviation Safety Engineering and Research
2871 Sky Harbor Blvd.
Phoenix, Arizona
A Division of
Flight Safety Foundation, Inc.

for
U. S. ARMY TRANSPORTATION RESEARCH COMMAND
FORT EUSTIS, VIRGINIA

by

William H. Reed
Donald F. Carroll

Approved:

James W. Turnbow, Ph. D.
Director of Engineering

Victor E. Rothe
Manager, AvSER Division
Flight Safety Foundation, Inc.

Merwyn A. Kraft
Research Coordinator
Flight Safety Foundation, Inc.

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## SUMMARY

This report presents a compilation of observations of airframe deformation which occurred during a full-scale droned crash of a CH-21A helicopter.

The information contained in this report, though general in nature, is intended to convey an understanding of the type and severity of damage which can be expected to occur in a moderately severe accident involving a CH-21A helicopter.

The report contains verbal descriptions of the crash sequence and the airframe deformations which resulted, as well as photographs of typical component damage.

## CONCLUSIONS

Based upon the information contained in this report, it is concluded that:

1. Although this crash was severe, the occupiable areas of the aircraft remained reasonably intact. With properly designed occupant restraint, incorporating energy absorption devices, this crash was potentially survivable.

2. When CH-21A aircraft are involved in accidents with circumstances similar to those encountered in this test, rupture of the main fuel cell is highly probable. This means that an extreme fire hazard exists in CH-21A aircraft involved in accidents of this nature.

3. In helicopter accidents which are similar to this test crash, the lower structure of the helicopter fuselage and any items rigidly attached to the lower structure will encounter vertical accelerations far above human tolerance levels.

4. The landing gear had little apparent effect on the downward motion of the aircraft.

## RECOMMENDATIONS

In order to produce helicopters which are more crashworthy, it is recommended that:

1. The magnitude and direction of airframe accelerations encountered in helicopter crashes and the known limits of human tolerance to impact acceleration be given more consideration in the design of restraint systems for helicopter use.

2. More care be exercised in the design and placement of fuel tanks so as to minimize the problem of postcrash fire.

3. Landing gear for helicopters be designed to absorb large quantities of energy under crash conditions. It would be preferable to absorb the energy by decelerating the helicopter under moderate acceleration through a relatively long stroke.

# INTRODUCTION

In order to improve the crashworthiness of rotary-wing aircraft, it is necessary to obtain more technical data concerning the behavior of aircraft of this type under actual crash conditions. Ideally, the data should be obtained during an actual crash of a full-scale aircraft, under controlled conditions. Accordingly, a long-range program, progressively leading to full-scale droned crash tests of rotary-wing aircraft, has been initiated by the U. S. Army Transportation Research Command in conjunction with the Flight Safety Foundation.

This report is a compilation of observations of airframe deformation which occurred during such a full-scale droned crash test of a CH-21A helicopter on 12 September 1962. Several experiments related to crash injury research were conducted during this test. The results of these investigations are contained on other reports.*

The objective of this test was to produce crash conditions which simulated a severe but potentially survivable accident in which the conditions at impact were near the upper limits of survivability as determined by structural collapse of the helicopter fuselage. Review of the test data reveals that these conditions were simulated during the subject test.

* References 2, 3, 5, 7, and 8 contain more detailed information concerning the various individual experiments conducted during this test.

## DESCRIPTION OF THE TEST ITEM

The test item consisted of a CH-21A helicopter (Figures 1 and 2). Seven instrumented anthropomorphic dummies were positioned on various crew seats, troop seats, and litters, at various locations within the cockpit and the cargo compartment. The left cockpit crew seat (copilot's seat) was removed from the aircraft to provide space for the installation of the remote guidance mechanism. Both cargo compartment doors were removed to allow sunlight to enter the fuselage as an aid to photography.* In addition, aircraft equipment which was not necessary for the completion of the experiments was removed to save weight and to eliminate the possibility of this equipment interfering with the experiments. The gross weight of the aircraft at the time of the crash was 12,300 pounds. Maximum allowable gross weight of the CH-21A is 14,500 pounds.

To reduce the fire hazard, fuel was drained from the main fuel cell and replaced with 200 gallons of water so that impact pressures would be produced within the tank similar to those which would be encountered in a tank containing gasoline. The water was dyed red for source identification and to show the pattern made by spilled fuel. Fuel for engine operation was carried in a specially mounted auxiliary tank located at the tail of the aircraft on the upper surface of the horizontal stabilizer.

The aircraft exterior was given a coat of flat yellow paint with the exception of certain areas which were color coded to aid photographic identification. A red band was painted around the fuselage just aft of the cargo compartment to be used as a focusing point for ground cameras.

The interior of the aircraft was given a spray coat of flat white paint to provide more light for the high-speed cameras and to provide a neutral background for contrast with the color-coded test equipment and other interior items being studied.

* The cargo compartment doors of the CH-21A are nonstructural. Removal of the doors has no effect on the pattern of structural deformation.

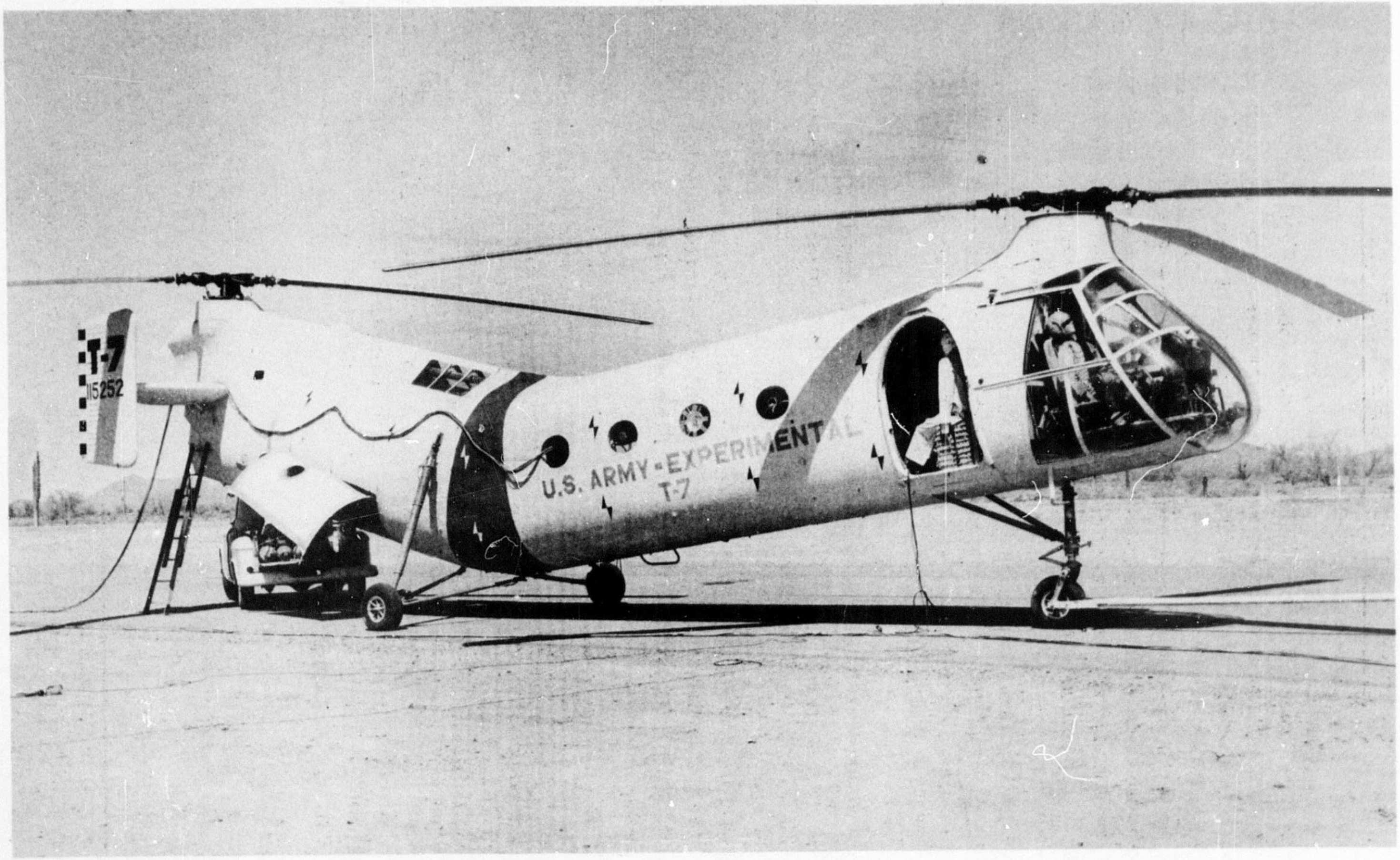


Figure 1. Test Vehicle Prior to Crash.

Note the various markings on the fuselage which aid in the study of structural deformation during crash.

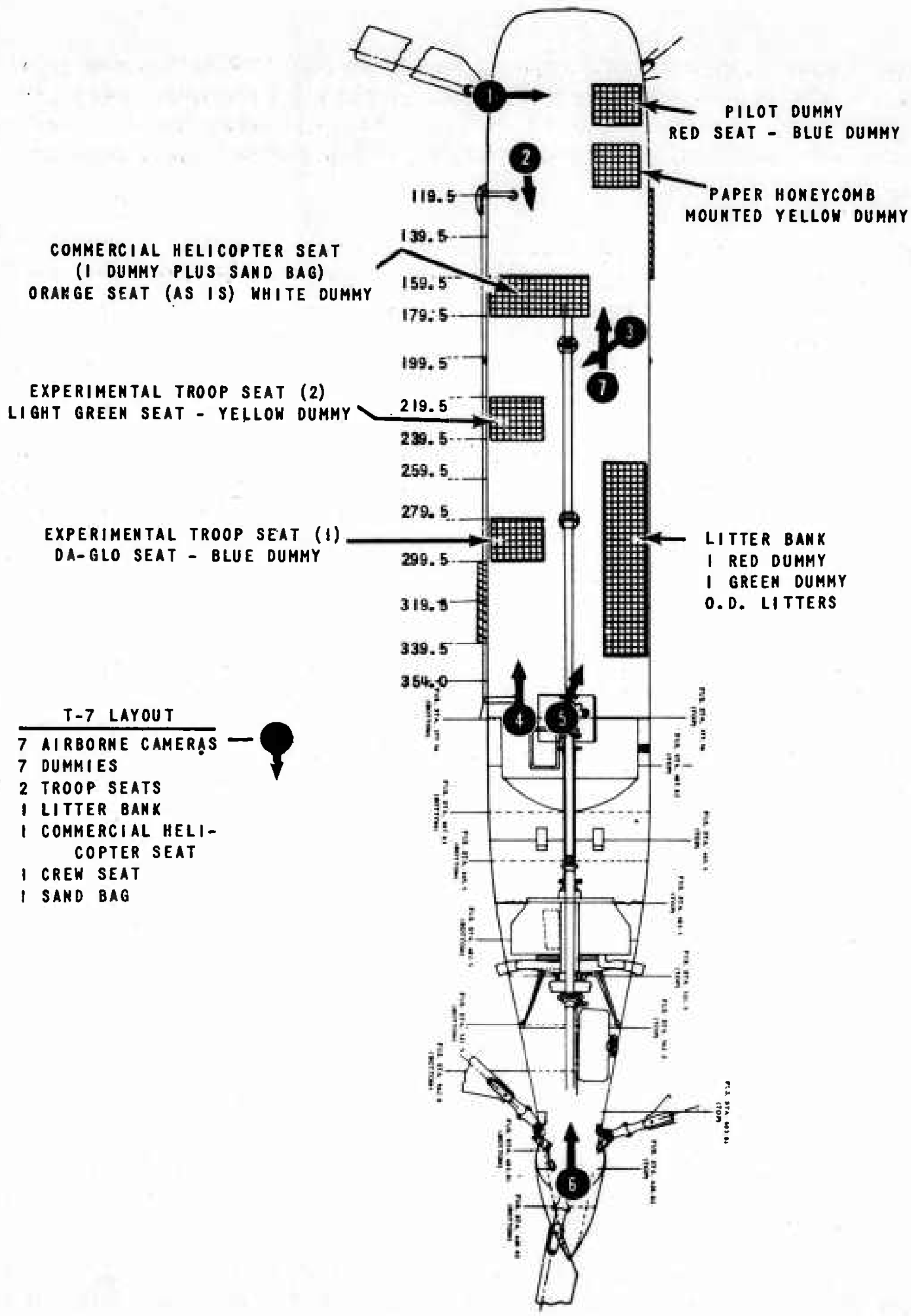


Figure 2. Layout of Equipment in Test Vehicle.

## EXPERIMENTAL PROCEDURE

Immediately prior to the crash, a qualified CH-21A helicopter pilot started the engine of the test vehicle, performed the necessary pre-flight checks, and engaged the rotors. The pilot then left the aircraft, and it was lifted off and flown along a predetermined flight path by remote control.

Data were obtained by photography, by the use of airborne accelerometers and strain gages, and by postcrash investigation. Seven high-speed motion picture cameras were placed aboard the aircraft to photograph the motion of experimental objects. The dynamic behavior of aircraft structure within the field of view of these cameras is also shown by these photographs.

Cameras were also located at ground sites chosen to show views of each side of the aircraft and quartering front and rear views. Color film was used for all motion picture photography. A Fairchild flight analyzer camera was used to aid in determination of precise conditions of altitude and velocity which were attained in the test.

Accelerometers were located on the cockpit floor at fuselage station 90, on the cargo compartment floor at fuselage station 270, and on the cargo compartment ceiling at fuselage station 240 to measure accelerations of the basic structure in three directions, parallel to the aircraft axes. Accelerometers were also mounted on the forward and mid transmission housings to measure the vertical accelerations at these points. A comparison of the acceleration data obtained from these sources helps in understanding how the fuselage structure transmits impact forces. The raw data obtained from these accelerometers are contained in Appendix I.

Data generated by airborne instrumentation were transmitted through an umbilical cable to recording equipment located on the ground near the impact point. Correlation of the several channels of recorded data and the photographic information was made possible by simultaneous recording of a 60-cycles-per-second sinusoidal timing trace on all oscillograph records and superposition of 60-cycles-per-second flashes from internal camera timing lights on film as it was exposed. A common power source was used for all timing purposes. A time datum was established by setting off flash bulbs, which were visible to all cameras, and simultaneously recording as one channel of oscillograph data the electrical impulse which fired the flash bulbs.

The observations of airframe deformation which are presented in this report are the result of postcrash investigation and detailed study of film from the high-speed cameras.

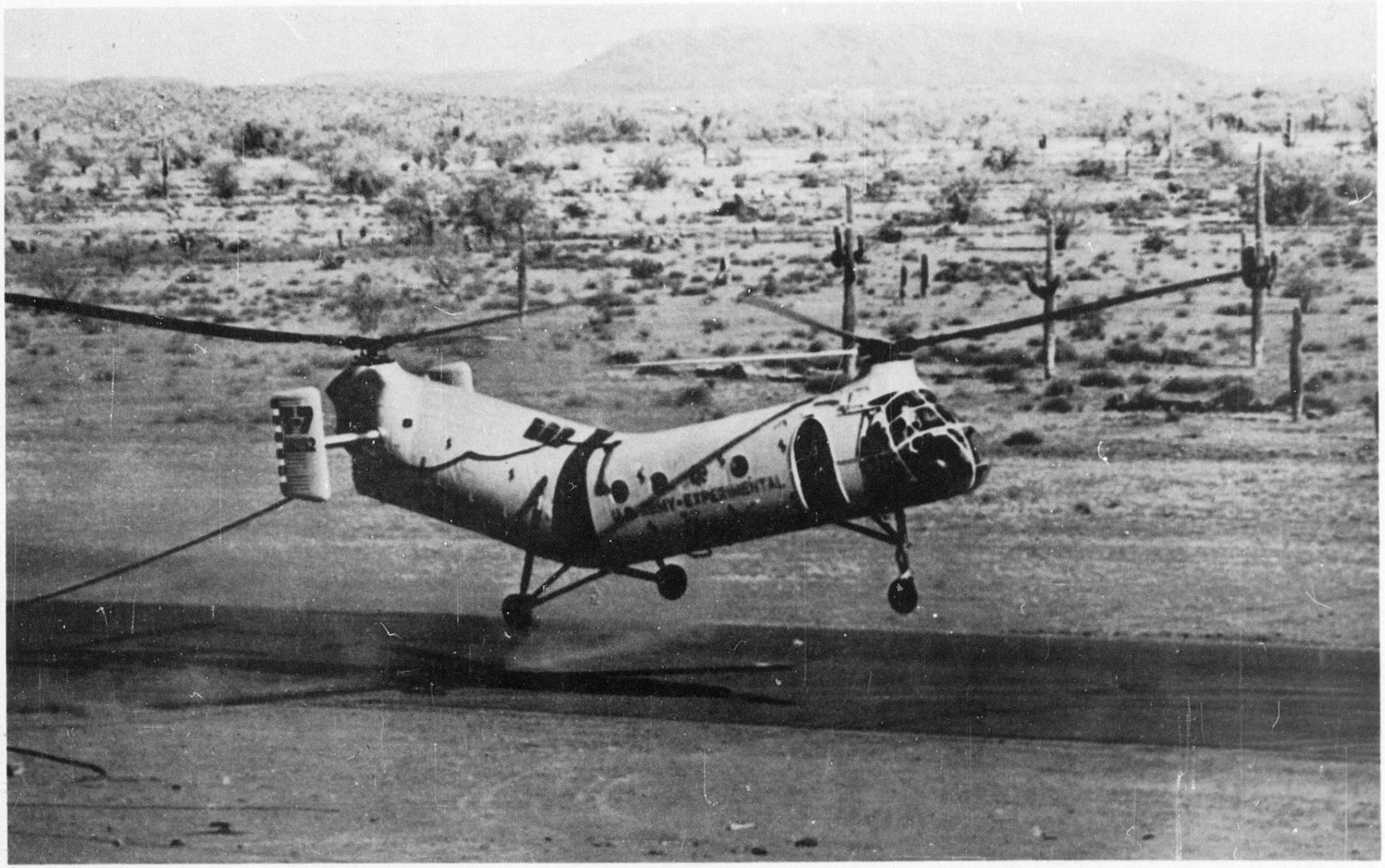

Figure 3. Test Vehicle Just After Lift-Off.
Note trailing umbilical cable through which all data were recorded during actual crash sequence.
(U. S. Army Photograph)

# EXPERIMENTAL RESULTS

## GENERAL

The flight conditions achieved up to the point of impact are set forth below and shown diagrammatically in Figure 4. This combination of conditions yielded a severe but realistic crash.

| | |
|---|---|
| Maximum altitude . . . . . . . . . . . . . . . . | 57 feet |
| Vertical velocity at impact . . . . . . . . . . | 40 feet per second |
| Horizontal velocity at impact . . . . . . . . . | 48 feet per second |
| Resultant flight path velocity . . . . . . . . . | 62.5 feet per second |
| Angle of flight path with horizontal runway at impact . . . . . . . . . . . . . . . . . . . | 40 degrees |
| Forward fuselage pitch angle at impact . . | 3 degrees nose up |
| Roll attitude at impact . . . . . . . . . . . | 4 degrees left roll |
| Yaw attitude at impact . . . . . . . . . . . . | Negligible |

Initial impact occurred on the helicopter nose wheel followed by contact on the main gear 0.052 second later. All landing gear collapsed immediately upon contact with the runway without apparent effect on the downward movement of the fuselage.

The fuselage struck the runway 0.124 second after initial impact. At impact the lower structure of the forward fuselage section was crushed and the aft fuselage section was bent downward sufficiently to contact the runway. The right side of the fuselage skin ruptured vertically just aft of fuselage station 360 as the aft section bent downward. The aft fuselage section then rebounded, as shown in the action sequences of Figures 5, 6, and 7, and twisted to the left. The main fuel tank ruptured during this sequence of events, and liquid was observed spilling from the main fuel tank into the cargo compartment of the aircraft 0.370 second after initial impact as the aircraft skidded along the runway.

The aircraft skidded some 43 feet from the point of initial impact. In the skid, the forward fuselage section rotated clockwise approximately 40 degrees, while the aft section rotated approximately 65 degrees* (Figure 8).

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* The magnitudes of the angles as stated here are accurate within plus or minus 5 degrees. This information is included primarily to aid the reader in visualizing the dynamic sequence of events which occurred during the crash.

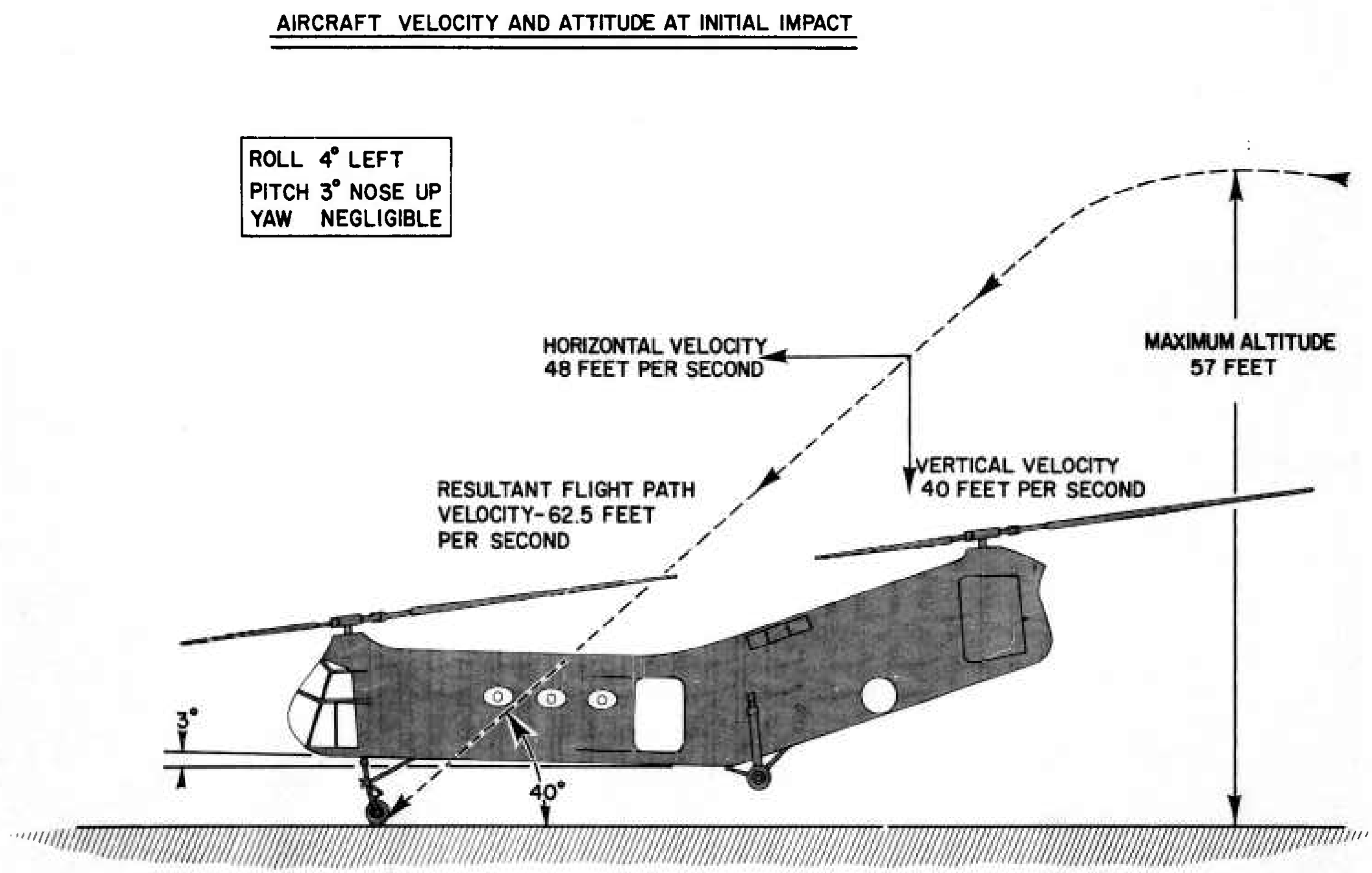


Figure 4. Crash Conditions of Altitude and Velocity.

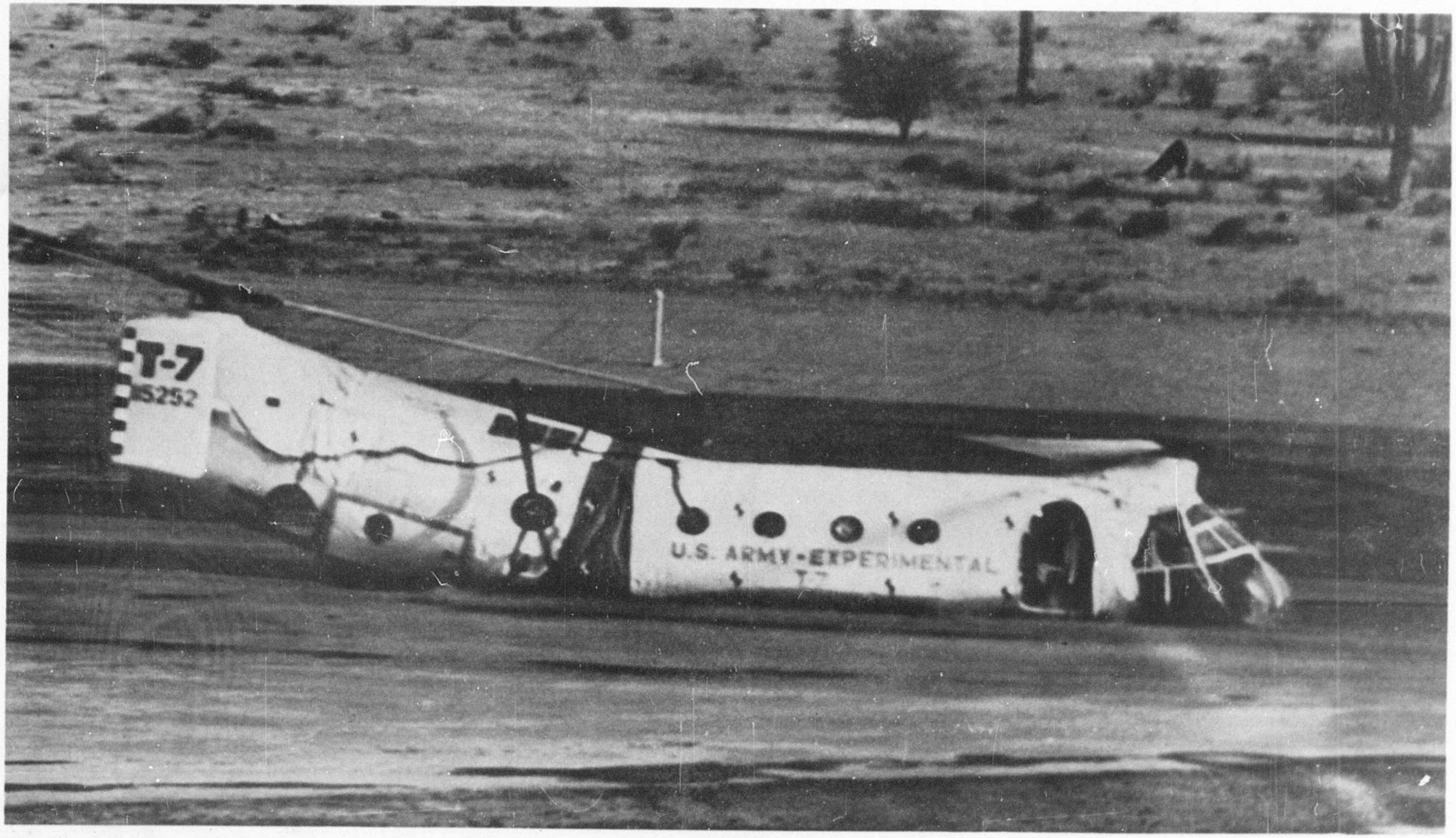


Figure 5. Test Vehicle 0.240 Second After Initial Impact.
Notice that the landing gear have failed, and that the lower fuselage is crushing due to impact. Also, notice the skin wrinkles in the dark band just forward of the main landing gear and in the tail section just above the engine compartment access door as well as the skin wrinkles below the forward rotor between the door and the cockpit. (U.S. Army Photograph)

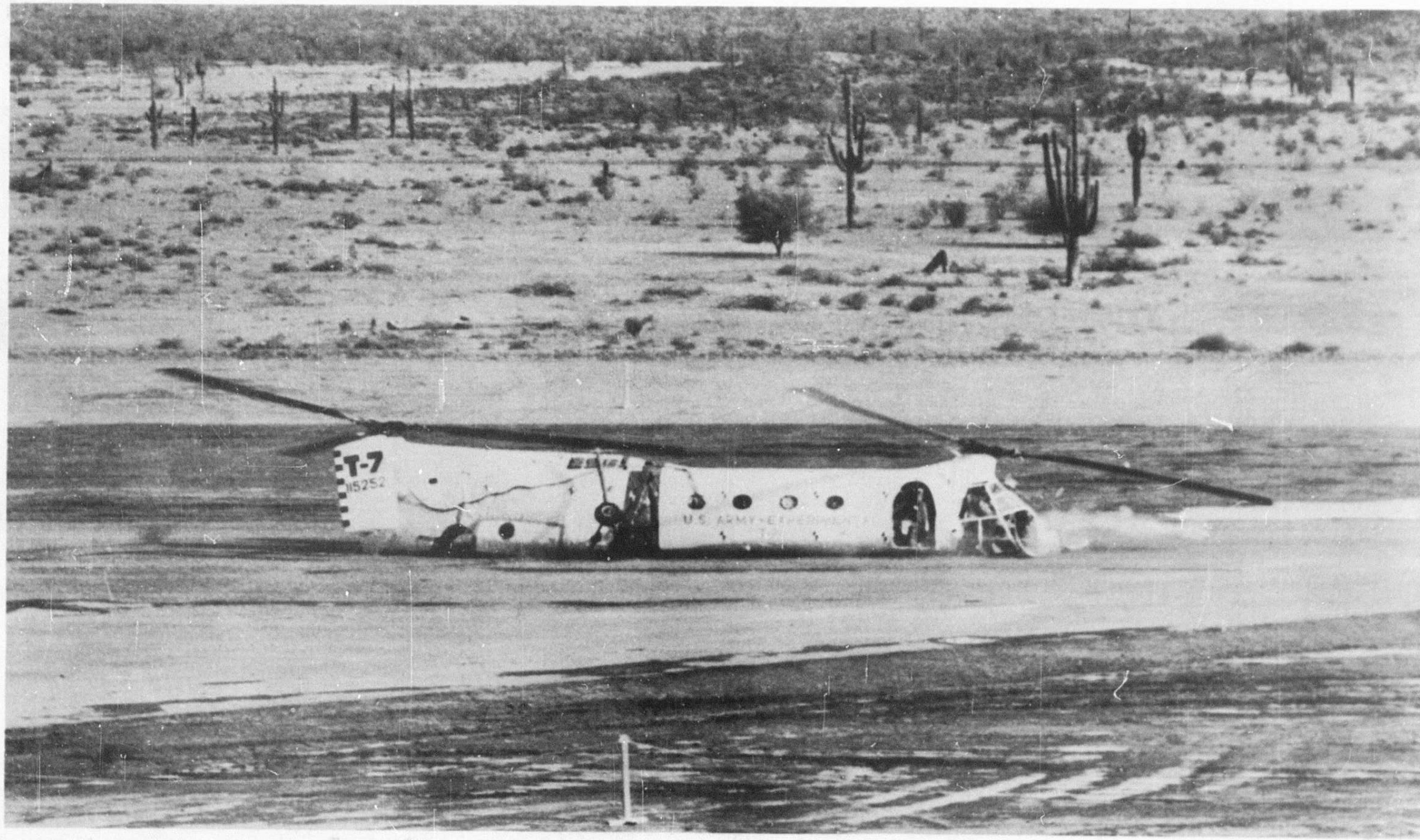


Figure 6. Test Vehicle 0.331 Second After Initial Impact.

Notice that the tail of the fuselage has bent downward until the entire bottom surface of the fuselage is in contact with the ground. It will also be noted that the fuselage is beginning to tear in the dark band just ahead of the main landing gear near the top of the fuselage. (U. S. Army Photograph)

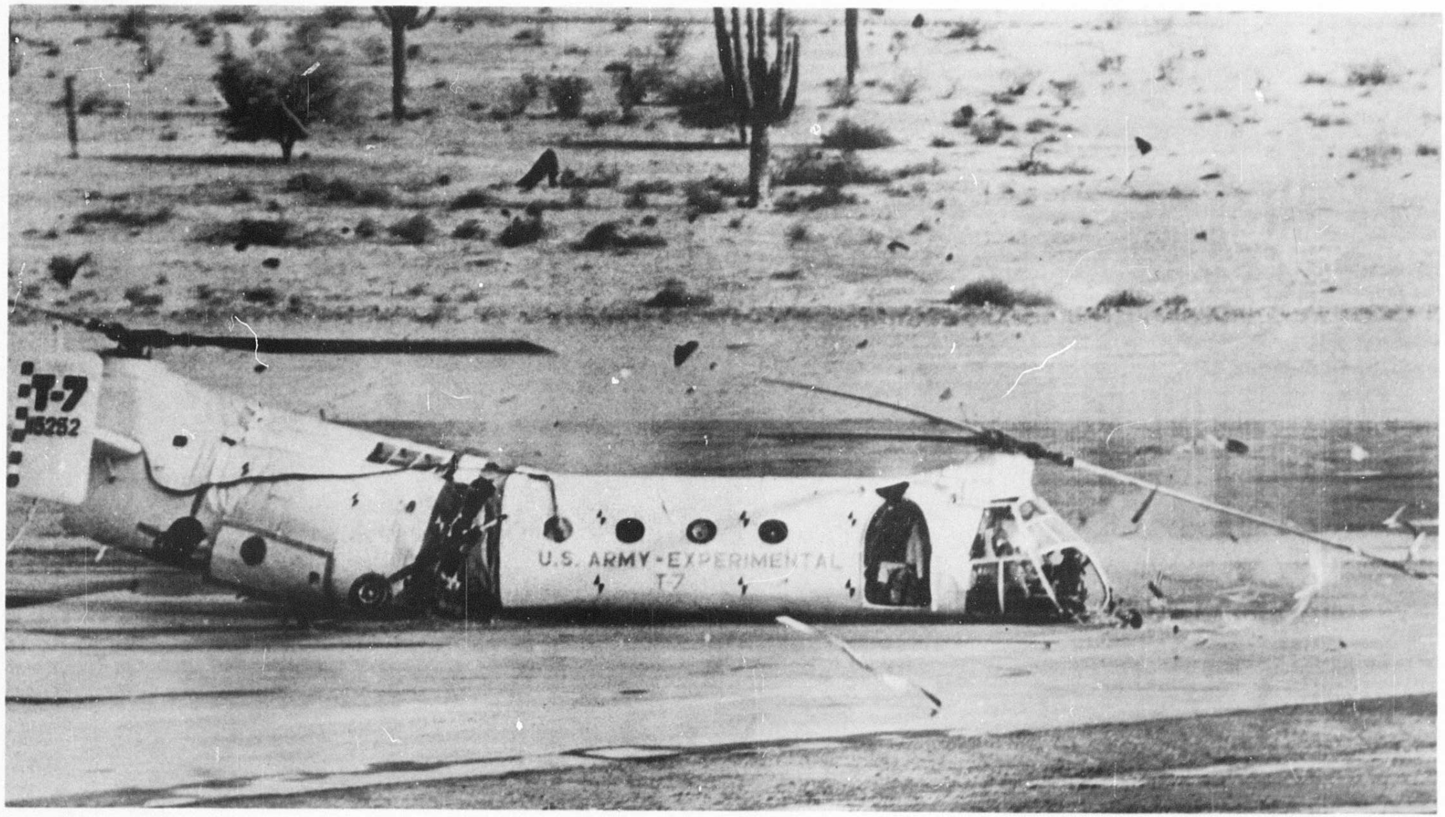


Figure 7. Test Vehicle 0.948 Second After Initial Impact.

The aircraft is bouncing and the tail is moving upward again toward its normal position. Notice that the rip in the dark band around the fuselage has opened considerably and liquid is spilling out through this opening onto the runway. Also, the forward rotor blades have now contacted the ground and are breaking up. (U. S. Army Photograph)

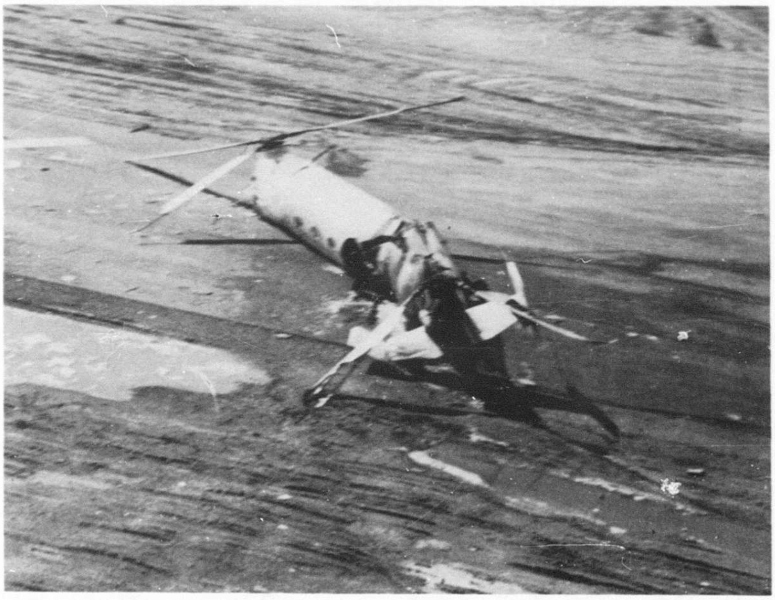

Figure 8. Attitude of Test Vehicle After the Crash.
Notice how the aircraft turned to the right and the tail section bent to the left.

A fire, which is explained fully in another report,[3] occurred during the crash. The fire did not consume any of the helicopter structure and had no effect on the pattern of structural deformation; thus, it will not be discussed further in this report.

Following are observations of damage to general areas and systems of the aircraft. These observations are generally the result of postcrash investigation correlated with data obtained from high-speed cameras.

## LANDING GEAR SYSTEM

The nose wheel broke away from the oleo strut, and then the nose landing gear oleo strut broke free from the aircraft at its attachment to the fuselage. As the aircraft settled to the ground, the nose wheel bounced up into the bottom of the fuselage, and the oleo strut assembly folded aft so that both were trapped beneath the forward section of the cargo compartment (Figures 9, 10, and 11).

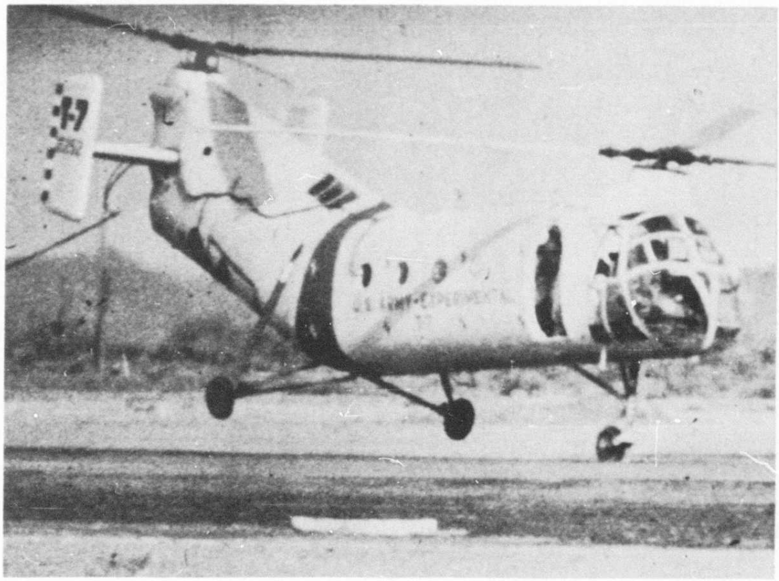

Figure 9. Sequence Photo of Nose Landing Gear Failure.

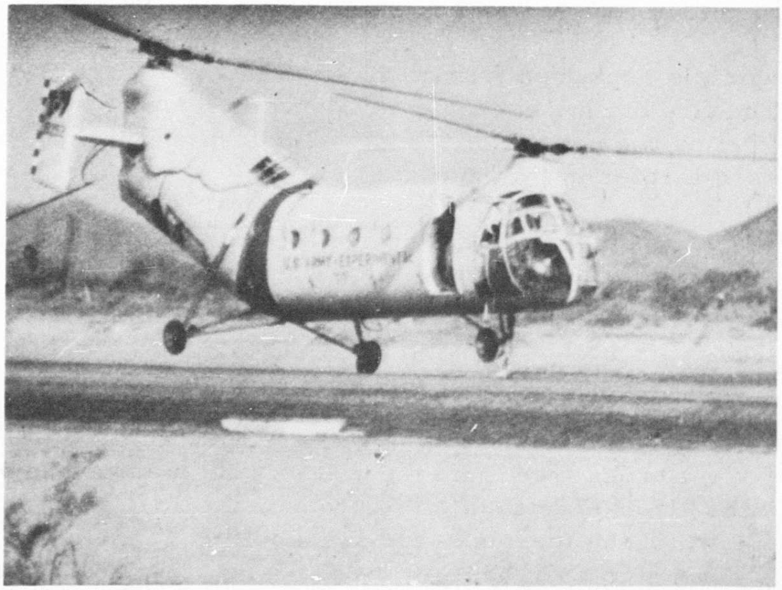

Figure 10. Sequence Photo of Nose Landing Gear Failure.

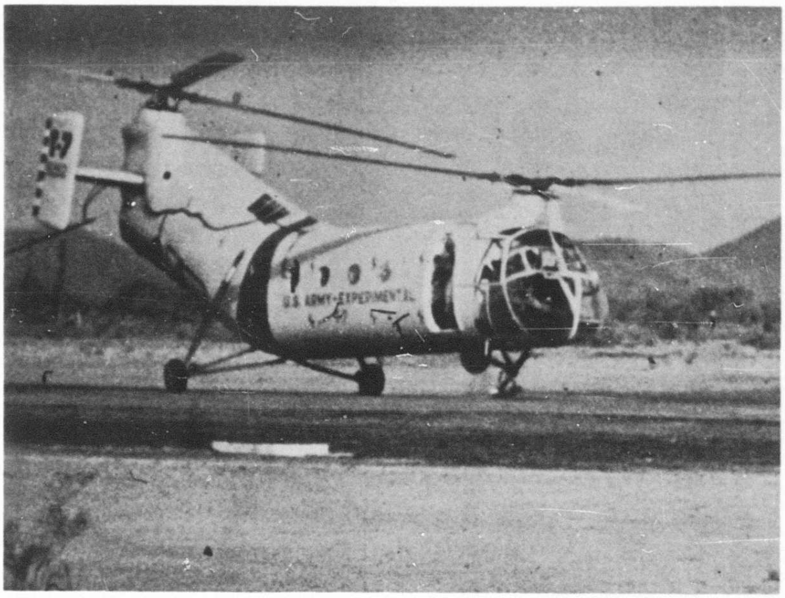


Figure 11. Sequence Photo of Nose Landing Gear Failure.

Both main landing gear support fittings, parts numbers 4251048-1 left side and 4251048-2 right side, failed (Figures 12 and 13). Study of high-speed motion picture film revealed that prior to the fitting failure, the main landing gear oleo struts were not compressed any appreciable amount; they acted as solid compression members. Impact forces were transmitted directly to the support fittings, causing their immediate failure. The oleo struts were then free to rotate about their lower attach point as the fuselage continued downward toward impact. The upper ends glanced off the fuselage skin and moved upward and outward relative to the support fittings without doing any major damage to the fuselage structure.

As the aircraft continued to settle and contacted the runway, the main landing gear V-brace assemblies punctured the lower fuselage, doing serious damage to the lower fuel tank structure and ripping the inverter from its mounts in the plenum chamber (Figures 14 and 15).

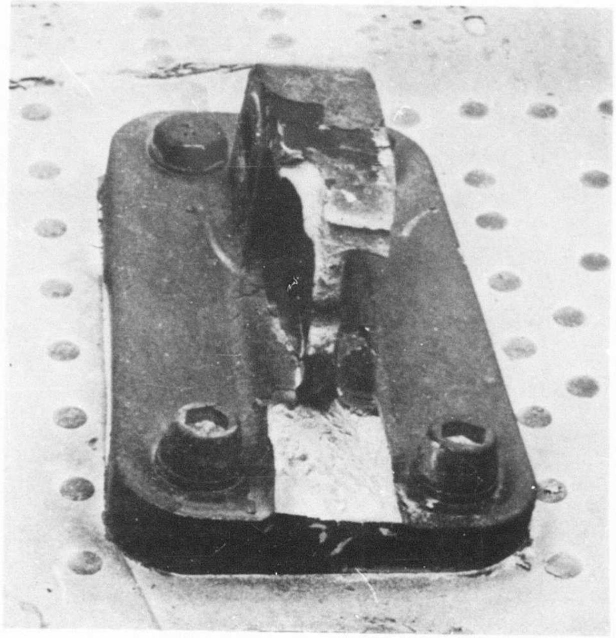

Figure 12. Main Landing Gear Support Fitting, Right Side.

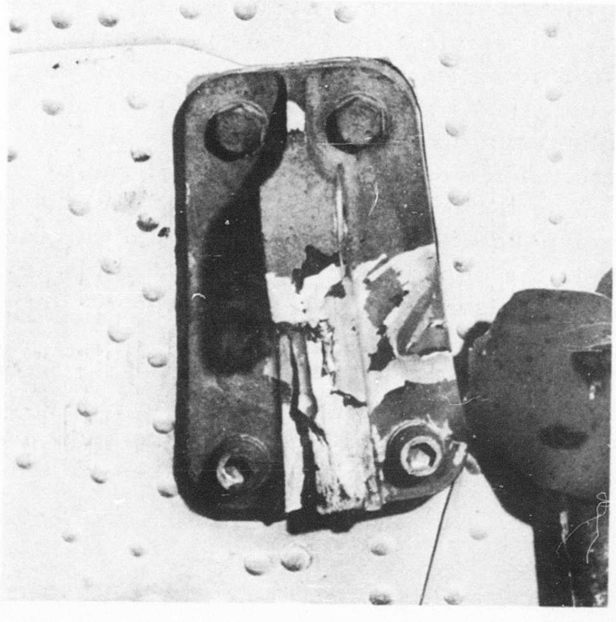

Figure 13. Main Landing Gear Support Fitting, Left Side.

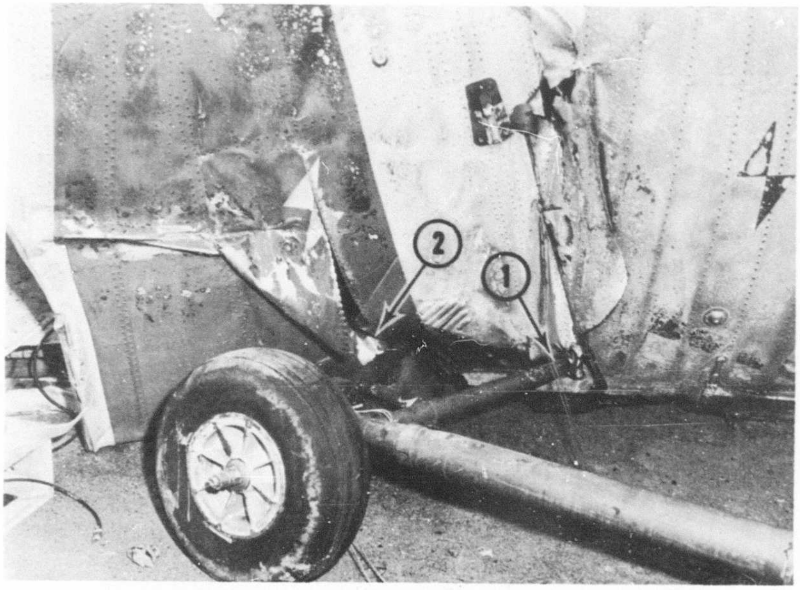


Figure 14. Left Main Landing Gear.
The rear V-brace (arrow 1) ripped through the fuselage and tore the inverter from its mounting in the plenum chamber. The forward V-brace (arrow 2) smashed upward into the main fuel cell.

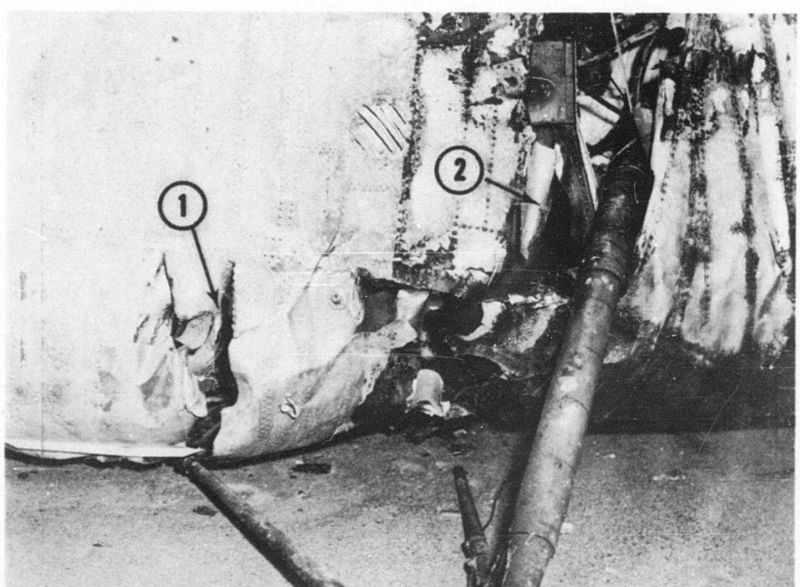


Figure 15. Right Main Landing Gear.
Notice the damage to the lower fuselage structure done by the V-brace assembly (arrow 1). Also notice the jagged hole in the main fuel cell just aft of the landing gear oleo strut (arrow 2).

## COCKPIT AREA

The primary structure in the cockpit area remained essentially intact. The floor sustained only minor damage, which was apparently caused by the failure of the pilot's seat and its subsequent impact with the floor. The main floor beams did not fracture when the fuselage substructure crushed, but permanent deformation did occur.

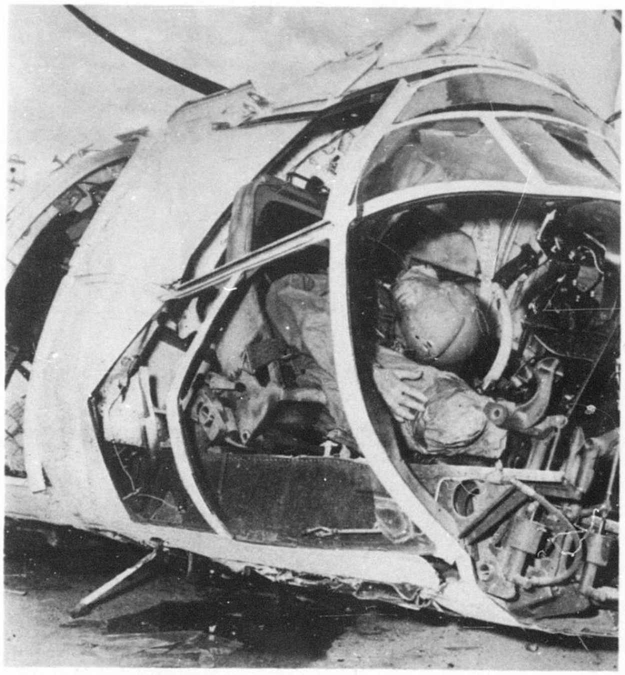

Figure 16. Right Side of Cockpit Following Crash.
Note the deformation of the window frame structure and the pilot's seat extending partly out one window. Also notice the window frame structure resting on the instrument console.

The bulkhead behind the pilot's seat, fuselage station 97.036, partially collapsed due to the high compressive load imposed upon it by the forward rotor transmission. The partial failure of this bulkhead allowed the transmission and rotor head to move downward and forward, causing collapse of the cockpit window frame structure, which then struck the instrument console (Figure 16). All the cockpit windows were broken. The partial failure of the station 97.036 bulkhead also resulted in buckling of fuselage skin aft of this station (Figure 17).

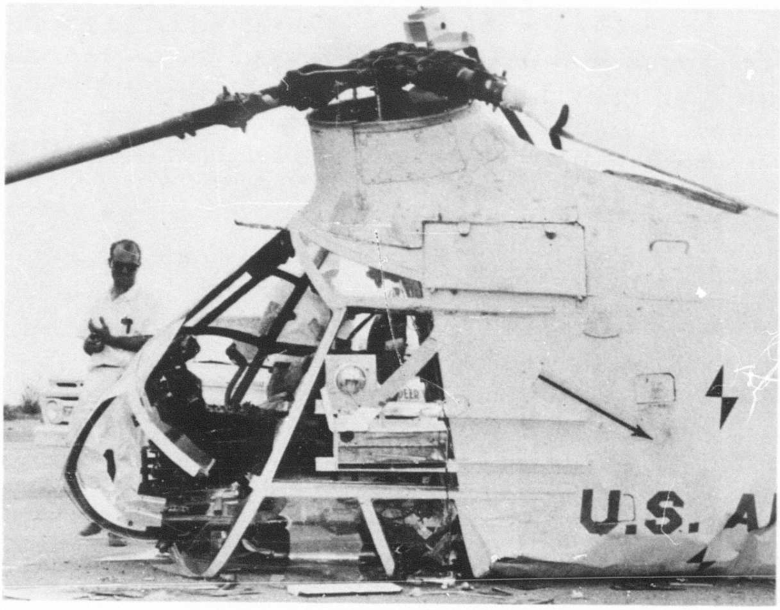


Figure 17. Left Side of Cockpit Following Crash.
Notice the deformation of window frame structure and the skin wrinkles aft of the cockpit (arrow).

The cockpit remained inhabitable throughout the crash, but because the cockpit is small and crowded with protruding objects, it is likely that both pilot and copilot would have sustained injuries.

The cargo compartment floor was buckled and displaced upward due to the crushing of the structure beneath the floor. The most severe buckling of the floor occurred between fuselage station 100 and fuselage station 160 due to the collapsed nose landing gear assembly, which was trapped beneath the fuselage in this area.

Fuselage structure above the normal troop seat attach points (approximately 17 inches above the normal floor line) remained almost completely intact on both sides of the aircraft. The relatively minor amount of structural deformation above this line is evidenced by the fact that plexiglass passenger compartment windows were not broken or popped out on impact. The forward rotor drive shaft remained in place, showing no evidence of impending failure at any of its attachments to the upper fuselage structure.

During the postcrash investigation, the escape hatches at the top of the cargo compartment were removed. No difficulty was encountered in their removal.

Figures 18 through 21 show the extent of fuselage deformation and cabin floor damage. It is impossible to predict the extent or severity of the injuries, but it is likely that cockpit occupants in this crash would have needed assistance to evacuate the aircraft quickly.

## CARGO COMPARTMENT

The entire cargo compartment lower body, including fuselage skin, floor support structure, and lower sections of body frames, was crushed by impact. Because the aircraft impacted with a slight left roll (Figure 4), the lower fuselage sustained more damage on the left side than on the right side.

The forward displacement of the forward rotor transmission and rotor head assembly caused the failure of a row of fasteners at the junction of the fuselage station 119.50 bulkhead and the upper fuselage skin at the aft edge of the right side cowling assembly, part number 2258020-7. The tearout of this row of fasteners progressed outboard into a tear of the fuselage skin panel immediately outboard of the cowling assembly. The upper fuselage skin aft of the fastener failure was buckled due to the tensile load which was applied by the forward displacement of the rotor head.

## AFT FUSELAGE SECTION AND FUEL TANK

When the aircraft contacted the runway, the aft fuselage section bent downward. The deflection was of such magnitude that the fuselage straightened out and the bottom surfaces of both the forward fuselage section and the aft fuselage section were in contact with the runway simultaneously. This bending of the fuselage resulted in failure of the fuselage structure and rupture of the fuselage skin between fuselage station 360 and fuselage station 407, just forward of the main landing gear.

The break in the fuselage extended from beneath the fuel tank on the bottom of the aircraft, up the right side, and over the top of the aircraft. The fuselage structure beneath the fuel tank and plenum chamber was badly mangled by a combination of impact, sliding, and puncturing by the main landing gear V-brace assemblies. The V-braces punctured the fuselage on both sides of the aircraft.

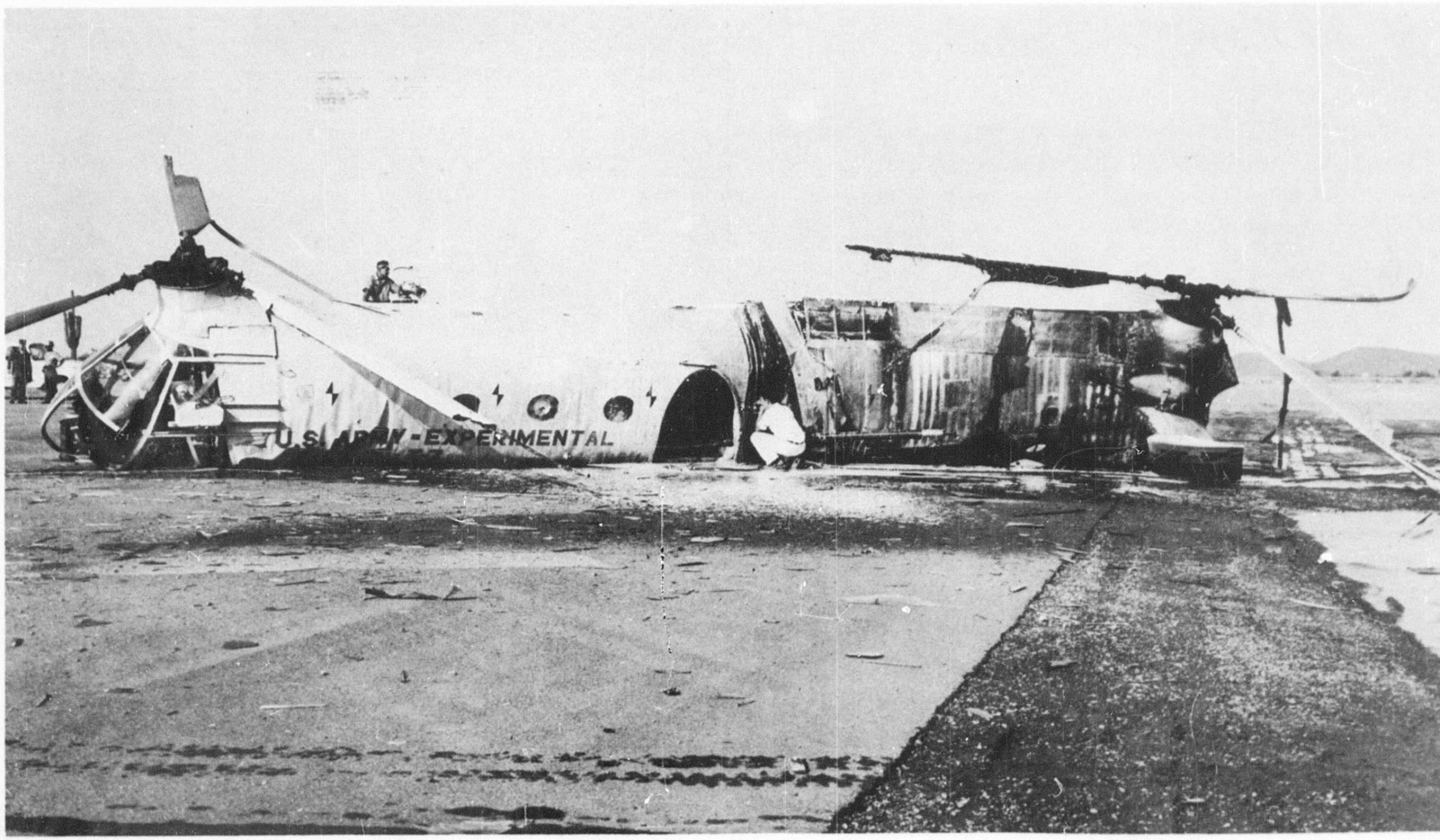


Figure 18. Overall View of Left Side of Aircraft Wreckage.


Figure 19. Overall View of Right Side of Aircraft Wreckage.

Notice that the lower fuselage structure is not damaged as badly on this side as it is on the left side, due to the slight roll which was present at impact.

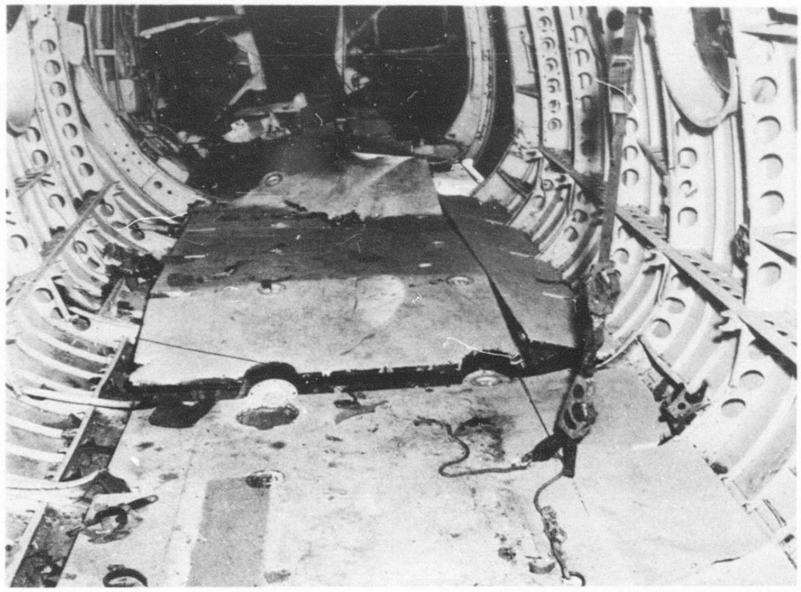

Figure 20. Overall View of Passenger Compartment Floor. This view is toward front of aircraft, after removal of experimental equipment.

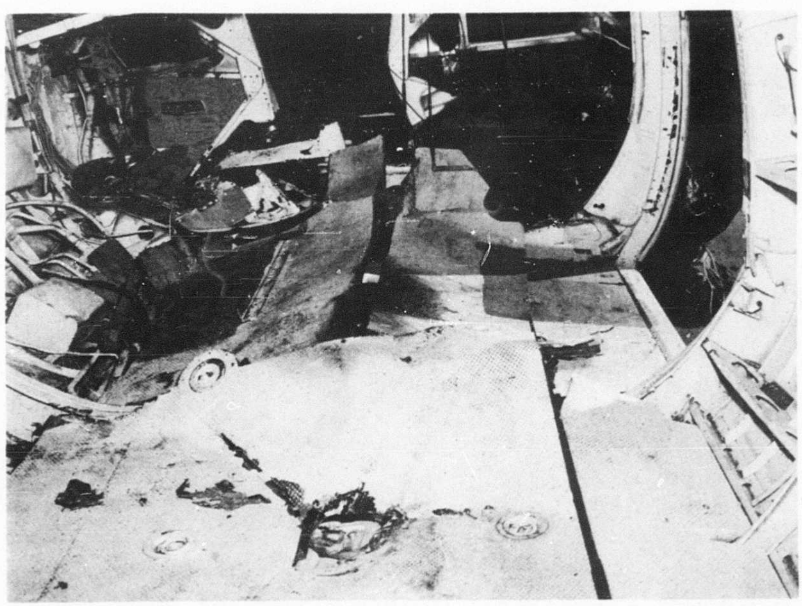

Figure 21. Severe Buckling of Passenger Compartment Floor, Fuselage Station 100 to Fuselage Station 160.
This deformation was caused by the collapsed nose landing gear which was trapped beneath the fuselage.

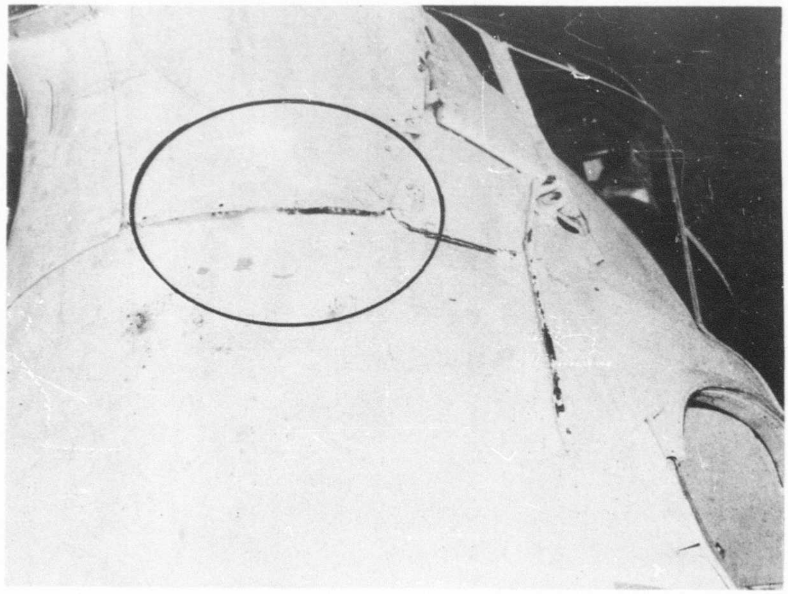

Figure 22. Upper Fuselage Aft of Forward Rotor Head. This photograph shows the separation of the cowling and the skin panel on the upper fuselage skin aft of the forward rotor head, fuselage station 119.50.

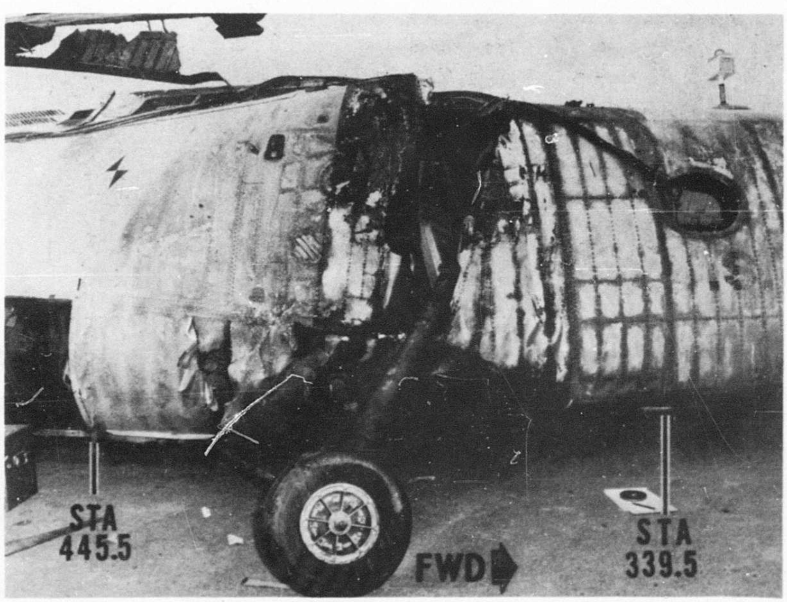


Figure 23. Fuselage Failure, Right Side.

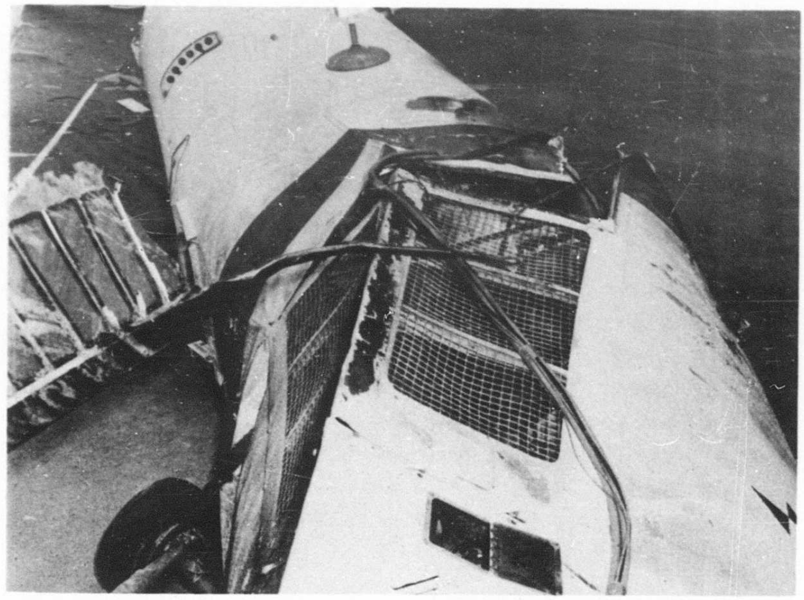

Figure 24. Fuselage Failure, Top View.
View looking forward showing the continuation of the fuselage rupture over the top of the fuselage. Notice the angular deflection of the entire tail section.

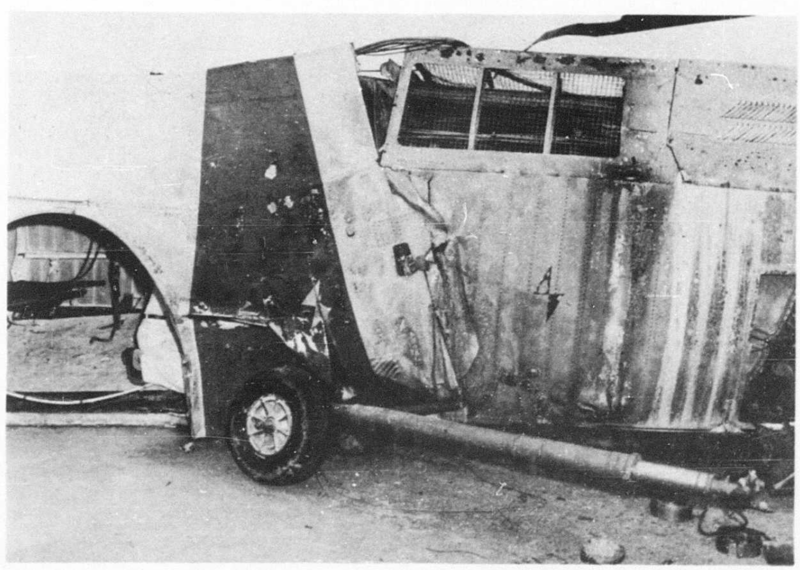

Figure 25. Fuselage Damage, Left Side, Opposite Fuselage Break.
This photograph shows the structural failures opposite the ruptured fuselage skin.

The firewall remained essentially intact, although it was buckled extensively. There were approximately 30 rivets missing midway up the right side of the firewall. The open holes left by the rivet failures and the slight cracks which appeared between the sheets of metal when the rivets failed provided sufficient space to allow fuel and fuel vapors to reach the plenum chamber and subsequently reach the engine, creating a potential fire hazard.

The fuel tank structure was extensively damaged. The top of the tank was the only part which remained intact. Figure 15 shows the beginning of a tear at the upper right side of the front of the tank. This tear extended all the way around under the fuel tank at the forward edge, to the top of the left side. The left side of the tank and the front of the tank were buckled due to the crushing of the bottom of the aircraft by impact. The bottom of the tank was torn and punctured to such an extent that it was virtually destroyed.

The entire tail section of the aircraft, aft of the break in the fuselage, came to rest bent to the left and twisted counterclockwise (viewed forward). The left vertical stabilizer contacted the ground and was consequently deformed, while the right side of the tail, which did not contact the ground, suffered little damage. When the tail bent down and to the left, both the engine drive shaft and the aft rotor drive shaft were pulled out of the center rotor transmission, which remained attached to the forward fuselage section. The engine remained attached to its engine mounts; the engine mounts and engine compartment structure maintained their structural integrity. Therefore, the engine was adequately protected from crash damage which might have added to the fire hazard.

# EVALUATION OF TEST RESULTS

The conditions achieved in this test closely simulated a moderately severe accident condition. The airframe deformations which occurred during this test are representative of the damage which can be expected when aCH-21A helicopter is involved in an accident under similar conditions.

Under other impact conditions, the damage pattern might not be the same as that encountered in this test. However, under moderate impact conditions, rupture of the fuel tank and extensive crushing of the structure which contacts the ground can be expected, as well as breaking of the fuselage where the tail section sweeps upward near the main landing gear.

The airframe accelerations encountered in this test are contained in Appendix I. Validity of the data is assured by instrument calibration immediately prior to the crash and immediately following the crash.

Study of high-speed motion picture film shows that the main landing gear contacted the runway 0.052 second after initial nose gear contact and that lower fuselage structure first contacted the runway 0.124 second after initial impact. Examination of the cargo compartment floor vertical accelerometer trace shows oscillatory accelerations increasing in magnitude until approximately 0.06 second after initial impact, then decreasing slowly in magnitude until approximately 0.125 second after initial impact, at which time the strongest acceleration pulses occurred, lasting until approximately 0.250 second after nose wheel contact.

The cargo compartment floor vertical accelerometer data were integrated to obtain the vertical velocity versus time curve of Figure 26. This curve shows a vertical velocity change at the end of 0.250 second of approximately 40.75 feet per second, which correlates closely with the vertical velocity of 40 feet per second which was obtained from the Fairchild Flight Analyzer data.

Integration of the vertical velocity curve yields a vertical displacement curve. The vertical velocity curve was integrated for the time period between initial impact and 0.176 second after initial impact, when the vertical velocity first became zero. This computation indicates that the section of the cargo compartment floor to which the accelerometer was attached moved downward a total of 66.85 inches after the nose wheel contacted the runway. The curve shows that during the time between nose wheel contact and main gear contact the vertical displacement was 24 inches, that between main gear contact and fuselage contact the dis-

placement was 29.75 inches, and that after the fuselage contacted the runway the floor continued downward another 13.10 inches.

With a CH-21A in the same attitude as the crash attitude, the vertical distance between the bottom of the nose gear and the bottom of the main gear is approximately 24.5 inches. The distance between the bottom of the main gear and the bottom of the fuselage at the point where the fuselage contacted the ground is approximately 29.0 inches, and the vertical distance between the bottom of the fuselage and the cargo compartment floor at the location of the accelerometer, fuselage station 270, is approximately 18 inches. Thus, the total possible vertical displacement of this accelerometer was approximately 70.5 inches. The difference between the computed vertical displacement and this maximum measured value is 4.65 inches. Postcrash observation placed the floor of the fuselage approximately 5 inches above the runway at the location of the accelerometer mount. The close correlation of both velocity change information and vertical displacement data as obtained from two separate sources is added proof of the validity of this accelerometer data.

In each of the experimental crashes of helicopters which AvSER has conducted, the helicopter airframe has been subjected to accelerations which exceed the known limits of human tolerance in the vertical direction and either exceed or border the limits in the horizontal direction. The implication of this is that if helicopter occupants are to be properly protected from the impact accelerations which occur in moderately severe crashes, they must be provided restraint systems which limit the maximum acceleration loads and prevent injurious contact between occupants and surroundings.

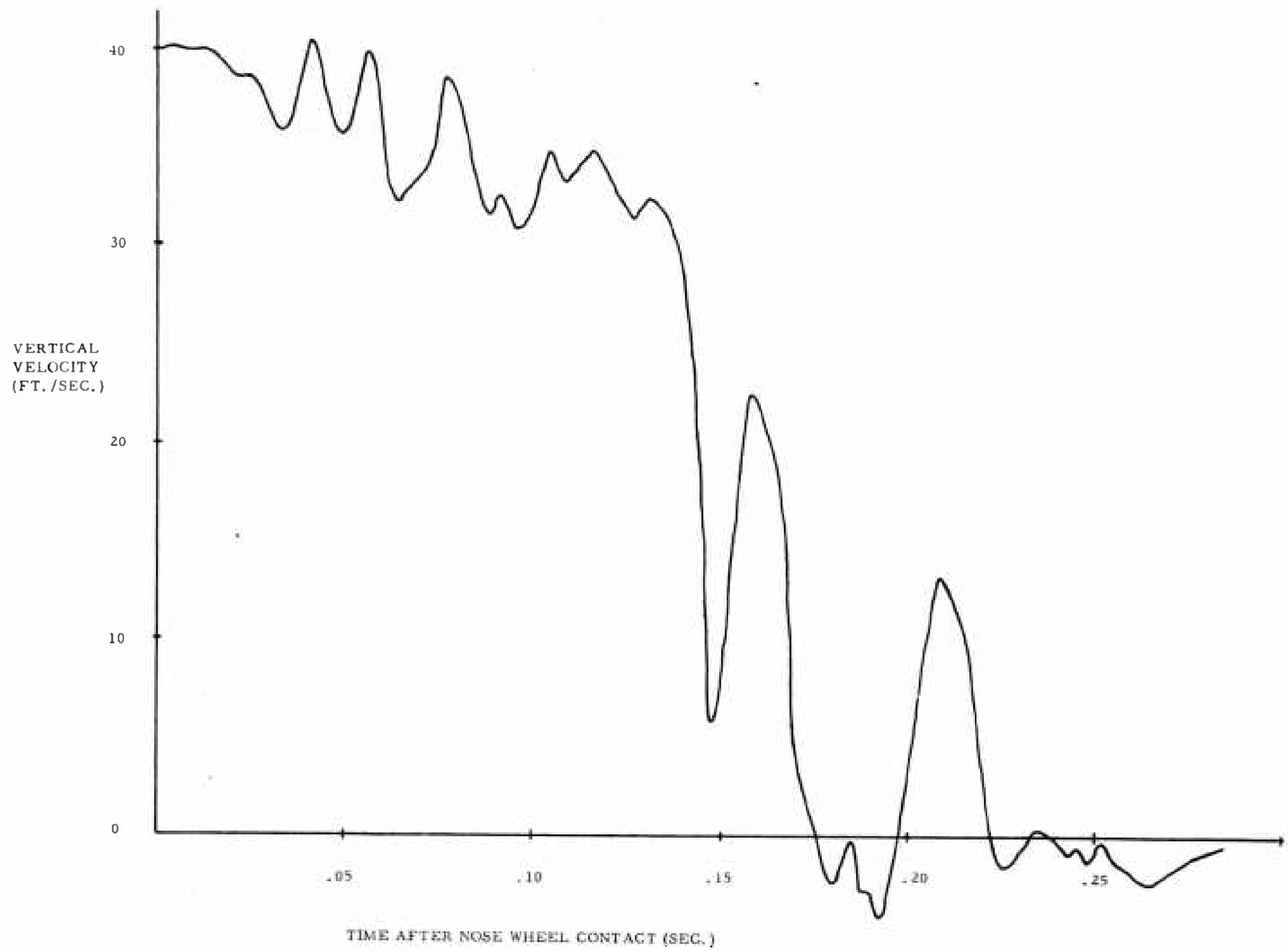


Figure 26. Vertical Velocity vs. Time, Cargo Compartment Floor.

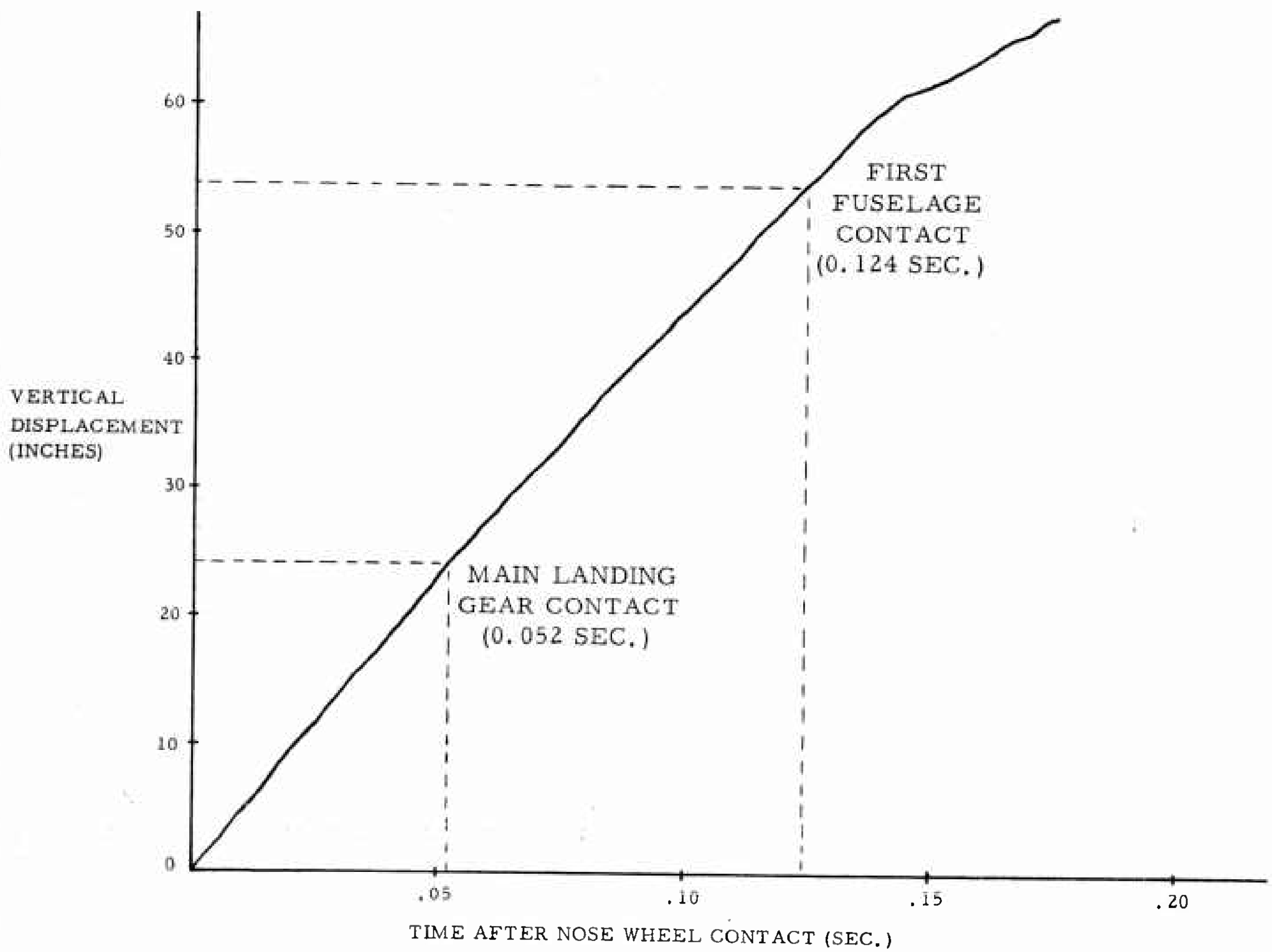

Figure 27. Vertical Displacement vs. Time, Cargo Compartment Floor.

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# APPENDIX I
# AIRFRAME ACCELEROMETER RECORDS

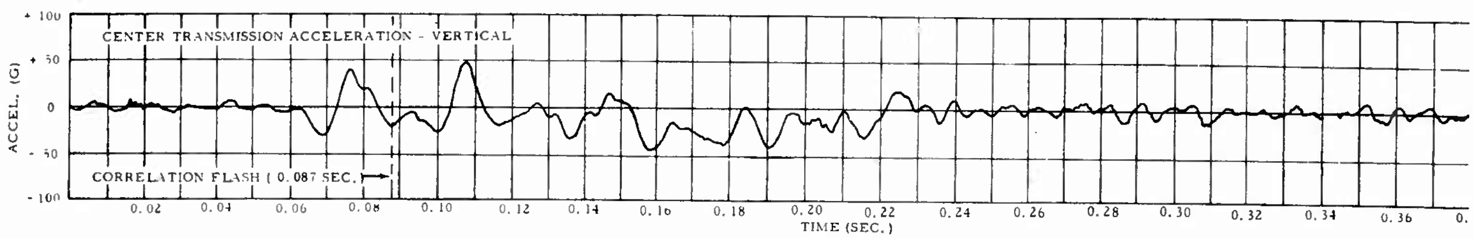
The accelerometer records obtained from instruments mounted directly to the airframe of the helicopter are shown on the following pages.

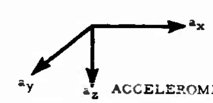
These curves conform to the aircraft computer standard sign convention.

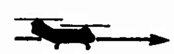
The times shown on these records indicate the elapsed time, in seconds, following initial impact. The time of initial impact, relative to the time of flash bulb firing, was determined from study of high-speed movies.

Only the portions of the curves from initial impact until 0.38 second after initial impact are shown here. Although the aircraft was still in motion, sliding along the runway, the most significant accelerations had already occurred by this time.

The usefulness of data obtained from the cockpit floor accelerometers is limited due to an unidentified external disturbance which occurred at approximately 0.18 second after initial impact, affecting all three accelerometer records.

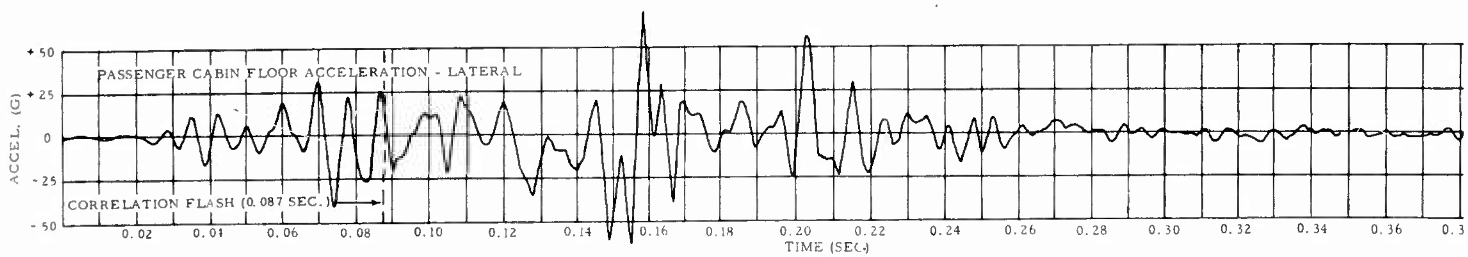


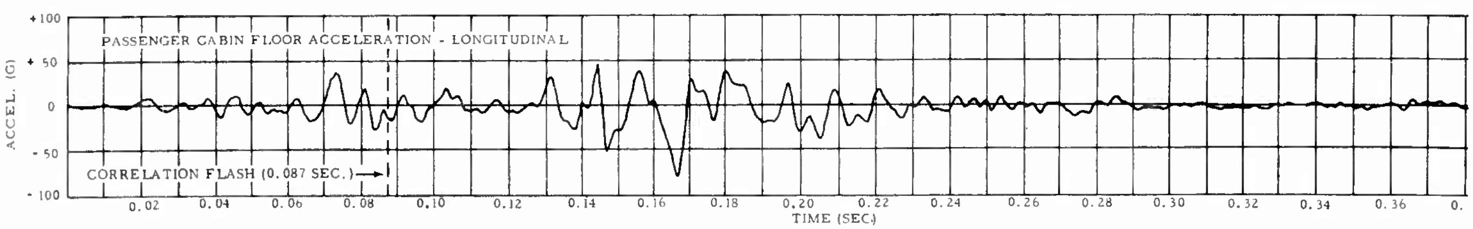



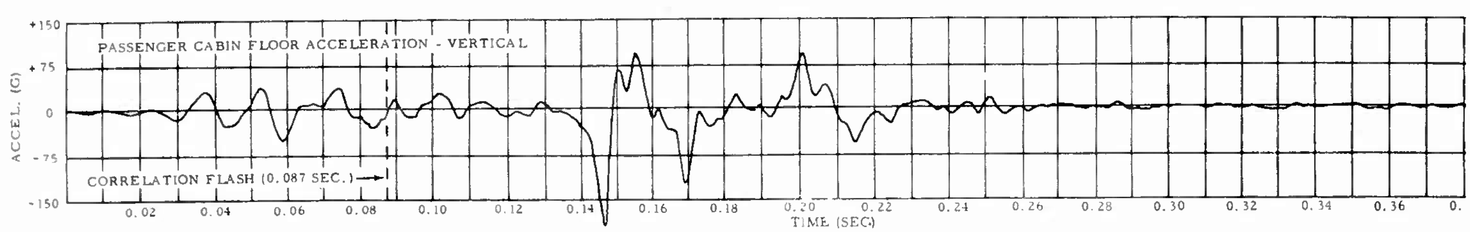



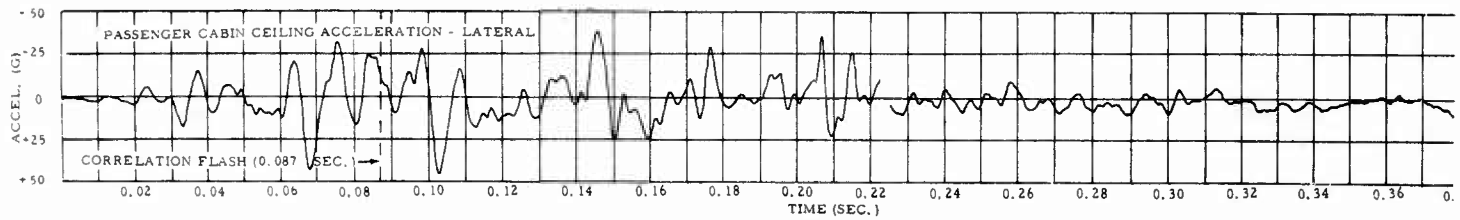
ACCELEROMETER SIGN CONVENTION

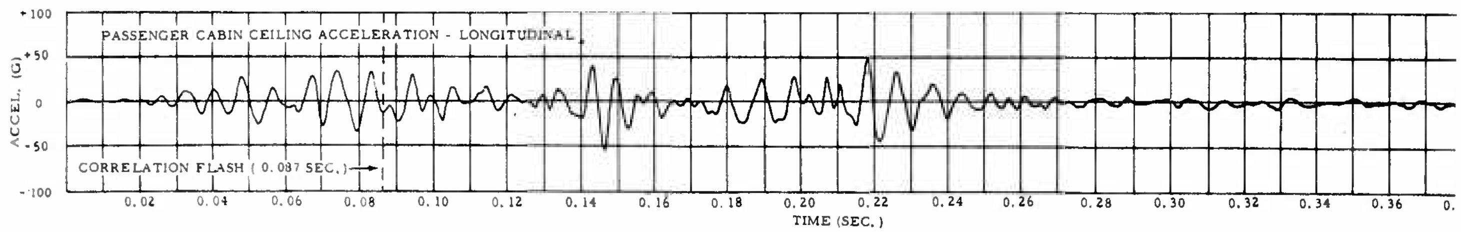
| LINEAR MOTION | AIRCRAFT COMPUTER STANDARD |
|---|---|
| FORWARD | $+a_x$ |
| BACKWARD | $-a_x$ |
| UPWARD | $-a_z$ |
| DOWNWARD | $+a_z$ |
| TO RIGHT | $+a_y$ |
| TO LEFT | $-a_y$ |

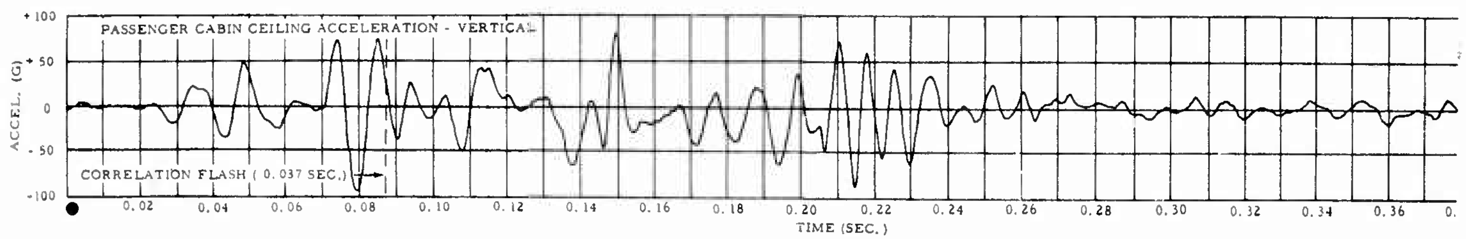

PASSENGER CABIN FLOOR ACCELERATION - LATERAL
CORRELATION FLASH (0.087 SEC.)
ACCEL. (G)
+50
+25
0
-25
-50
0.02
0.04
0.06
0.08
0.10
0.12
0.14
0.16
0.18
0.20
0.22
0.24
0.26
0.28
0.30
0.32
0.34
0.36
TIME (SEC.)


PASSENGER CABIN FLOOR ACCELERATION - LONGITUDINAL
CORRELATION FLASH (0.087 SEC.)
ACCEL. (G)
+100
+50
0
-50
-100
0.02
0.04
0.06
0.08
0.10
0.12
0.14
0.16
0.18
0.20
0.22
0.24
0.26
0.28
0.30
0.32
0.34
0.36
TIME (SEC.)


PASSENGER CABIN FLOOR ACCELERATION - VERTICAL
CORRELATION FLASH (0.087 SEC.)
ACCEL. (G)
+150
+75
0
-75
-150
0.02
0.04
0.06
0.08
0.10
0.12
0.14
0.16
0.18
0.20
0.22
0.24
0.26
0.28
0.30
0.32
0.34
0.36
TIME (SEC.)


PASSENGER CABIN CEILING ACCELERATION - LATERAL
CORRELATION FLASH (0.087 SEC.)
ACCEL. (G)
- 50
-25
0
+25
+ 50
0.02
0.04
0.06
0.08
0.10
0.12
0.14
0.16
0.18
0.20
0.22
0.24
0.26
0.28
0.30
0.32
0.34
0.36
TIME (SEC.)


PASSENGER CABIN CEILING ACCELERATION - LONGITUDINAL
CORRELATION FLASH ( 0.087 SEC.)
ACCEL. (G)
+ 100
+ 50
0
- 50
- 100
0.02
0.04
0.06
0.08
0.10
0.12
0.14
0.16
0.18
0.20
0.22
0.24
0.26
0.28
0.30
0.32
0.34
0.36
TIME (SEC.)


PASSENGER CABIN CEILING ACCELERATION - VERTICAL
CORRELATION FLASH ( 0.037 SEC.)
ACCEL. (G)
+ 100
+ 50
0
- 50
-100
0.02
0.04
0.06
0.08
0.10
0.12
0.14
0.16
0.18
0.20
0.22
0.24
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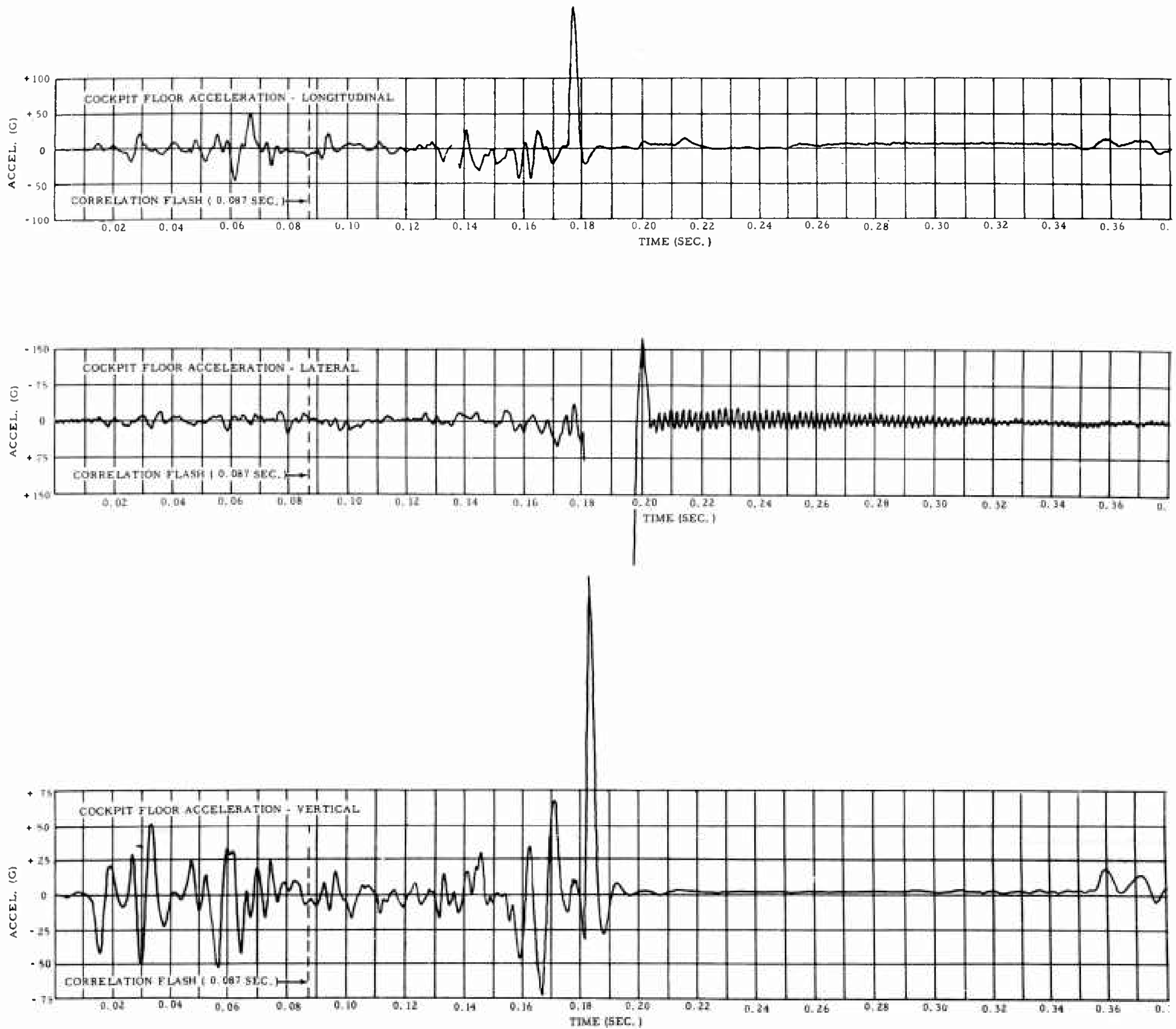
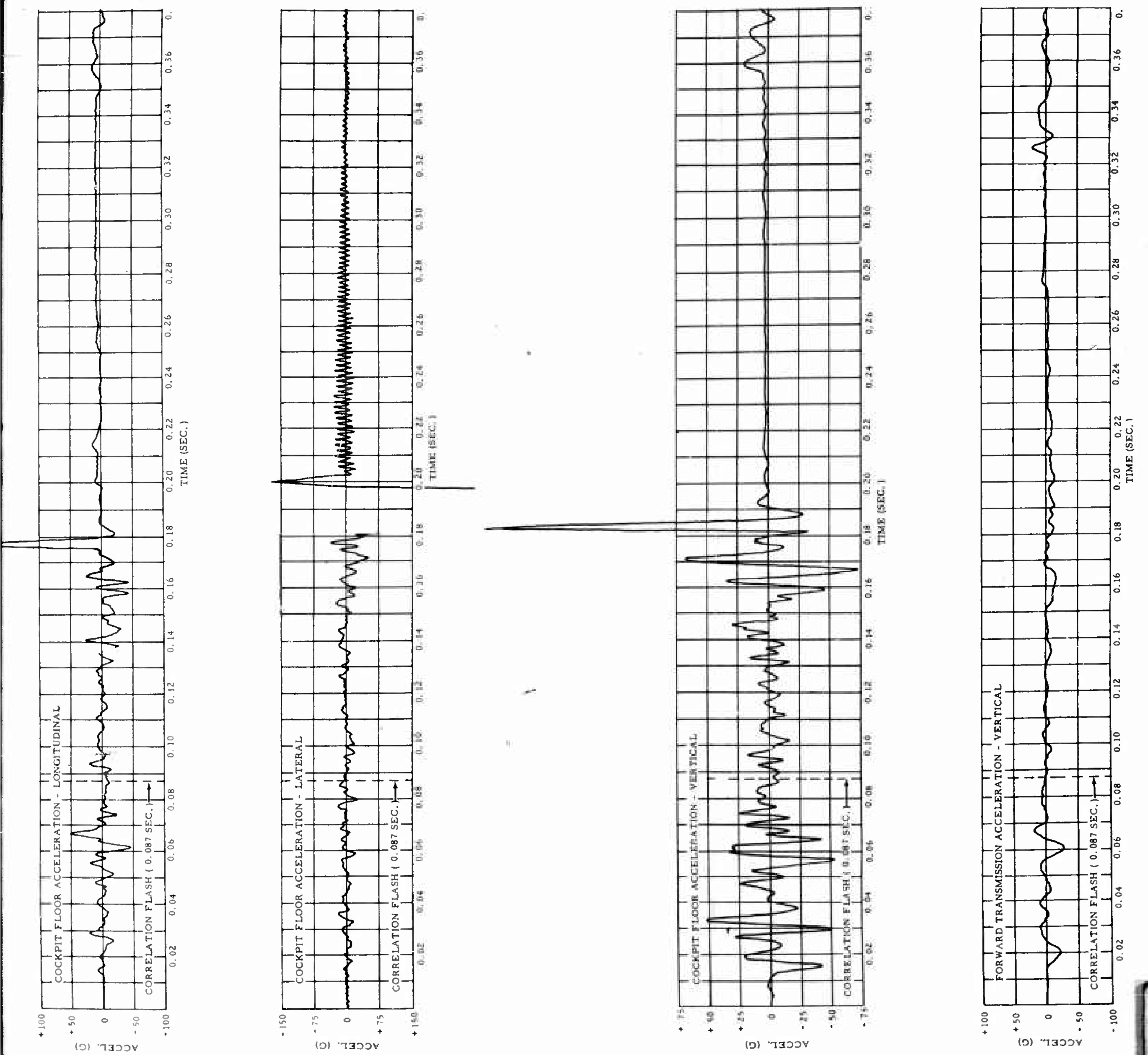
0.28
0.30
0.32
0.34
0.36
TIME (SEC.)


COCKPIT FLOOR ACCELERATION - LONGITUDINAL
CORRELATION FLASH ( 0.087 SEC. )
ACCEL. (G)
+100
+50
0
-50
-100
0.02 0.04 0.06 0.08 0.10 0.12 0.14 0.16 0.18 0.20 0.22 0.24 0.26 0.28 0.30 0.32 0.34 0.36
TIME (SEC.)
COCKPIT FLOOR ACCELERATION - LATERAL
CORRELATION FLASH ( 0.087 SEC. )
ACCEL. (G)
-150
-75
0
+75
+150
0.02 0.04 0.06 0.08 0.10 0.12 0.14 0.16 0.18 0.20 0.22 0.24 0.26 0.28 0.30 0.32 0.34 0.36
TIME (SEC.)
COCKPIT FLOOR ACCELERATION - VERTICAL
CORRELATION FLASH ( 0.087 SEC. )
ACCEL. (G)
+75
+50
+25
0
-25
-50
-75
0.02 0.04 0.06 0.08 0.10 0.12 0.14 0.16 0.18 0.20 0.22 0.24 0.26 0.28 0.30 0.32 0.34 0.36
TIME (SEC.)


COCKPIT FLOOR ACCELERATION - LONGITUDINAL
CORRELATION FLASH ( 0.087 SEC.)
ACCEL. (G)
TIME (SEC.)
COCKPIT FLOOR ACCELERATION - LATERAL
CORRELATION FLASH ( 0.087 SEC.)
ACCEL. (G)
TIME (SEC.)
COCKPIT FLOOR ACCELERATION - VERTICAL
CORRELATION FLASH ( 0.087 SEC.)
ACCEL. (G)
TIME (SEC.)
FORWARD TRANSMISSION ACCELERATION - VERTICAL
CORRELATION FLASH ( 0.087 SEC.)
ACCEL. (G)
TIME (SEC.)

## APPENDIX II
## INSTRUMENTATION DATA LIST

| Item | Use | Description |
|---|---|---|
| Electrical accelerometers | Acceleration sensing | Statham A5A-50-350 and A5A-100-350. |
| Tensionmeters | Force sensing | AvSER strain gage force links, 1000-pound and 2500-pound. |
| Recording oscillograph | Amplitude vs. time records of transducer outputs | CEC Model 5-114; 26 channel recording oscillograph with associated power supplies. |
| Photographic/ oscillograph data correlation device | Zero time datum for oscillographic and film data | Photo flash bulbs mounted in field of view of cameras. Firing pulse to bulbs recorded on oscillograph record for correlation. |

## APPENDIX III
## CAMERA DATA LIST

| | | |
|---|---|---|
| High-speed motion picture camera | Displacement vs. time for kinematics data | Photosonics 1B, 16 mm, high G tolerance 500 frames/second Ektachrome ER 430 film. |
| Normal-speed motion picture camera | General photographic coverage | 2 ea. Kodak 16 mm 64 frames/second; 2 ea. Bolex 16 mm 24 frames/second Kodachrome II Film. |
| Flight path analyzer | Horizontal and vertical velocity of test vehicle | Fairchild FDFA-044 flight analyzer. |
| Voltage generator | Correlation and timing of high-speed cameras | 115-volt AC generator, 60 cps timing pulse. |

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| Army Research Office, Durham | 2 |
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| U. S. Army Materiel Command Aviation Field Office | 2 |
| Bureau of Medicine and Surgery | 3 |
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| Armed Forces Institute of Pathology | 2 |
| U. S. Air Force Directorate of Flight Safety Research, Norton AFB | 1 |
| U. S. Army Board for Aviation Accident Research | 5 |
| U. S. Army Aviation Human Research Unit | 1 |
| U. S. Army Representative, U. S. Naval Aviation Safety Center | 1 |
| U. S. Naval Aviation Safety Center | 2 |
| Naval Air Materiel Center | 3 |
| Naval Air Development Center | 1 |
| Wright Development Division, Wright-Patterson AFB | 4 |
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| Helicopter Utility Squadron TWO, Lakehurst | 2 |
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| U. S. Srike Command | 1 |
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| National Library of Medicine | 1 |

Aviation Safety Engineering and Research, Phoenix, Arizona, CH-21A HELICOPTER AIRFRAME DEFORMATION UNDER A DYNAMIC CRASH CONDITION, W. H. Reed, D. F. Carroll, TRECOM Technical Report 63-77, January 1964, 43 pp., - illus. (Contract DA 44-177-AMC-888(T), USATRECOM Task 1A024701A12101. (Unclassified Report)

An obsolete U. S. Army CH-21A helicopter was remotely controlled to a crash designed to simulate a known accident configuration. (cont'd.)

UNCLASSIFIED

1. Helicopter Airframe Deformation After Crash
2. Contract DA 44-177-AMC-888(T)

Aviation Safety Engineering and Research, Phoenix, Arizona, CH-21A HELICOPTER AIRFRAME DEFORMATION UNDER A DYNAMIC CRASH CONDITION, W. H. Reed, D. F. Carroll, TRECOM Technical Report 63-77. January 1964, 43 pp., - illus. (Contract DA 44-177-AMC-888(T), USATRECOM Task 1A024701A12101. (Unclassified Report)

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